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RESEARCH MEMORANDUM

PRELIMINARY ANALYSIS OF HYDROGEN-RICH HYPERSONIC RAMJET OPERATION

By Roland Breitwieser and James F. Morris

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PRELIMINARY ANALYSIS OF HYDROGEN-RICH HYPERSONIC RAMJET OPERATION

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SUMMARY

Net thrust, fuel flow, and related performance indices were calculated for hydrogen-rich operation of nacelle-type and submerged ramjet engines at Mach numbers from 5 to 20. Hydrogen-air ratios considered were high enough to limit the combustor temperature to a maximum of 2000° K. This propulsion method can produce fuel specific impulses considerably greater than those of rockets if impulses are averaged over the above range of flight speeds.

Some trends of the effect of operating conditions on performance were determined; however, no optimums were defined, since the report is of a preliminary nature. For this reason, problems such as high hydrogen flow rates, use of this low-density cryogenic fluid, large exhaust-nozzle areas, and vehicle structure and design were not treated.

INTRODUCTION

The application of unique combustion cycles might result in air-breathing engines that can propel vehicles to very high flight speeds, speeds currently obtained only with rocket engines. In this report a modified ramjet cycle has been analyzed to determine its suitability for acceleration from a Mach number of 5 to 20. The particular cycle discussed combines diffusion by a normal shock at the inlet and addition of hydrogen at many times the stoichiometric rate in the combustor. The hope is that the high impulse of ramjets at moderate flight speeds can be sustained over the entire speed range.

The cycle selected is intended to overcome three main obstacles to successful use of a ramjet over a range of very high speeds.

The first of these problems is matching inlet and exhaust geometries. Conventionally, large variations in inlet and exhaust areas and area ratios are required to cover wide variations in speed. The matching problem is met in this study by selecting a normal-shock inlet diffuser for use in the Mach 5 to 20 range. This fixed-geometry inlet stays on design if the proper back pressure is supplied. Addition of hydrogen to the high-temperature dissociated gas in the combustor to provide this back pressure was explored. At Mach numbers greater than 5, the normal-shock diffuser provides workable pressure ratios (diffuser pressure to ambient pressure). Also, the normal-shock diffuser may be suitable for use in high-speed aircraft like the configuration described in reference 1. These aircraft have a large base area that permits incorporating the engine in the fuselage. In this case the large cowl drags normally associated with normal-shock inlets can be absorbed by the fuselage.

The second problem is the effect of high stagnation energies on structure, and the third problem is the effect of dissociation on thrust. The kinetic energy of the incoming air is about 9000 Btu per pound at a Mach number of 20. If this energy were reflected in temperature (ideal gas, ratio of specific heats = 1.4), the stagnation temperature would be approximately 40,000° F. At a Mach number of 20 a real gas dissociates long before temperatures of 40,000° F are reached. The new species absorb some of the inlet energy in their formation. The lower temperature thus resulting cannot be relied upon to alleviate heat-transfer and structural problems, since much of the dissociated gas can recombine and release energy on or near the engine surfaces. Even more energetic gases exist in the combustor, where the enthalpy of the fuel has been added to the system. The problems in building an engine structure to contain the high-energy gas stream are obviously severe. If the dissociated gases do not recombine before or during the expansion process, the energy that produced the dissociation is not available for thrust.

Addition of hydrogen in quantities many times stoichiometric lowers temperatures to the point where simple engine structures can be used. The resulting low temperatures also recombine the dissociated air and combustion products. Although very high hydrogen flows are required to produce low combustion temperatures at the very high flight speeds under discussion here, these large hydrogen flows may also be required to cool the aircraft structure externally. In fact, the hydrogen temperatures selected for this analysis are based on the idea that hydrogen will be used for cooling prior to its use in the engine.

The analysis presented in this report should be considered exploratory. Primarily, the analysis attempts to describe the cycle and to determine the parameters that have the greatest effect on engine performance. The analysis was thus limited to a few selected cases. A flight path was selected to give a combustor pressure of 2 atmospheres. For this flight path the free-stream dynamic pressure is nearly constant at 1.08 atmospheres. Hydrogen (fuel) temperatures of 150°, 500°, and 850° K were used. Combustor-exit temperatures of 1200°, 1600°, and 2000° K were assigned. Fuel specific impulse, thrust coefficient, and exhaust-nozzle areas were computed for complete expansion of exhaust gases. A single case of partial expansion was also treated. The results are presented for two methods of installing engines on an aircraft: (1) a submerged engine and (2) a nacelle engine; the results differ only in the assignment of external body forces.

SYMBOLS

A	area per unit of weight-flow rate of air
C_F	net thrust coefficient
F_n	net thrust per unit weight-flow rate of air
f	fuel-air ratio
g	gravitational constant
H_T^o	enthalpy per mole at temperature T and pressure of 1 atm
I	fuel specific impulse, net thrust per unit of weight-flow rate of fuel
J	constant for conversion of work units to heat units
M	Mach number
m	exhaust-nozzle-exit stream momentum per unit of weight-flow rate of air, $(1 + f)V_e/g$
p	static pressure
q	dynamic pressure
R	universal gas constant
S_T^o	entropy per mole at temperature T and pressure of 1 atm
T	static temperature
V	velocity
w	weight flow
γ	ratio of specific heats
φ	equivalence ratio (actual fuel-air ratio divided by stoichiometric fuel-air ratio)

Subscripts:

a	air
c	combustor conditions (zero velocity)

e	nozzle-exit conditions
f	fuel
H_2	hydrogen
H_2O	water
N_2	nitrogen
O_2	oxygen
0	free-stream or ambient

ANALYSIS

The modified ramjet cycle is based on the use of hydrogen at fuel-air ratios in excess of stoichiometric so that combustion temperature does not exceed 2000° K. Since high hydrogen flows are involved at Mach numbers of 15 to 20 (approximately), it is of interest to consider what kind of thrusts may be expected from mass addition alone. Restricting this consideration to the high Mach number case and neglecting heat addition, the energy equation reduces to

$$w_a V_0^2 \approx (w_a + w_f) V_e^2$$

where w_a and w_f are weight flows of air and fuel, V_0 is flight velocity, and V_e is exit velocity of the jet. The various inlet and outlet enthalpy terms have been neglected because at high flight velocities $V_0^2/2g$ is much greater than the static enthalpies. From the preceding expression, $V_e = \sqrt{\frac{1}{1+f}}\, V_0$, where f is w_f/w_a; but, for complete expansion, net thrust is

$$\frac{F_n}{w_a} = (1 + f)\frac{V_e}{g} - \frac{V_0}{g}$$

$$\frac{F_n}{w_a} = \left(\sqrt{1 + f} - 1\right)\frac{V_0}{g}$$

The efficiency of this form of thrust production can be evaluated from fuel specific impulse. Fuel specific impulse I can be found from

$$I = \frac{F_n w_a}{w_a w_f} = \left(\frac{\sqrt{1 + f} - 1}{f}\right) \frac{V_0}{g}$$

In the limit as the fuel-air ratio goes to zero the quantity in parentheses approaches 1/2. The quantity is relatively insensitive to fuel-air ratio. For example, at $f = 1$ the value is 0.414.

Over a reasonable range of fuel-air ratios, the "fuel-rich" ramjet cycle may be approximated by $I \approx V_0/2g$. The approximation is crude, but it does indicate that interesting values of impulse may be attained at high velocities through simple mass addition.

The relations on thrust from mass addition given are not meant to imply that the proposed cycle is based on mass addition alone. At low Mach numbers the cycle is similar to a conventional ramjet cycle where most of the propulsive energy comes from burning the fuel. At these low Mach numbers only sufficient hydrogen is used to cope with the high-temperature dissociation and structure problem. At low Mach numbers, this cycle is similar to that discussed in reference 2. However, as flight speed is increased mass addition starts to contribute more to thrust. At all Mach numbers considered herein, energy added in the form of heat of combustion, and fuel enthalpy are important contributors to thrust.

In the initial review of materials for use in this cycle, hydrogen appeared to be the most promising. This selection was based upon heat of combustion, heat capacity, and stable structure of hydrogen at high temperatures.

Flight Conditions

The flight path was selected to yield a combustor pressure of 2 atmospheres with a normal-shock inlet; calculations were made for flight Mach numbers of 5, 10, 15, and 20. Ambient static conditions for these points, taken from reference 3, are as follows:

Flight Mach number	Ambient static pressure, atm	Ambient static temperature, $^{\circ}K$	Approximate altitude, ft	Dynamic pressure, atm
5	0.06124	211	63,500	1.072
10	.01548	229	93,000	1.082
15	.00689	242	111,000	1.085
20	.00388	257	125,000	1.087

Thermodynamic Data

Hydrogen gas (fuel) temperatures of 150°, 500°, and 850° K were chosen. Results were computed for complete expansion of exhaust gases from a combustor-exit pressure of 2 atmospheres and temperatures of 2000°, 1600°, and 1200° K. The maximum combustor temperature of 2000° K avoids the dissociation region and temperature conditions that produce severe structural problems. The lower temperatures of 1200° and 1600° K were introduced to give trends of performance with combustor temperature. The combustion products were assumed to be nitrogen, water, and hydrogen; this simplified calculations and caused an error of less than 1 percent in total enthalpy (ref. 2). Air was assigned the composition of 3.773 moles of nitrogen per mole of oxygen. Thermodynamic data for the molecular species were obtained from reference 4.

Calculation Methods

Flight Mach number and hydrogen and combustor-exit temperatures were selected. The following expression was then used to compute the equivalence ratio giving equal total energies for reactants and products:

$$\left(H_{T,0}^{o}\right)_{O_2} + 3.773\left(H_{T,0}^{o}\right)_{N_2} + 2\varphi\left(H_{T,f}^{o}\right)_{H_2} + 6.64T_0M_0^2$$

$$= 2\left(H_{T,c}^{o}\right)_{H_2O} + 3.773\left(H_{T,c}^{o}\right)_{N_2} + 2(\varphi - 1)\left(H_{T,c}^{o}\right)_{H_2}$$

Isentropic, frozen-composition expansion of combustion products was assumed, and the exhaust-nozzle-exit temperature was determined from the entropy relation,

$$\Delta S_T^o = \left[5.773 + 2(\varphi - 1)\right] R \ln \frac{p_e}{p_c}$$

$$= 3.773\left(S_{T,e}^o - S_{T,c}^o\right)_{N_2} + 2\left(S_{T,e}^o - S_{T,c}^o\right)_{H_2O} + 2(\varphi - 1)\left(S_{T,e}^o - S_{T,c}^o\right)_{H_2}$$

The enthalpy change for expansion was calculated from

$$\Delta H_T^o = 3.773\left(H_{T,e}^o - H_{T,c}^o\right)_{N_2} + 2\left(H_{T,e}^o - H_{T,c}^o\right)_{H_2O} + 2(\varphi - 1)\left(H_{T,e}^o - H_{T,c}^o\right)_{H_2}$$

The exhaust-nozzle-exit stream momentum and area were then computed:

$$m = (1 + 0.0293\varphi)\frac{V_e}{g}$$

$$= (1 + 0.0293\varphi)\sqrt{\frac{2J\left(-\Delta H_T^o\right)}{g[3.773(28.016) + 2(18,016) + 2(\varphi - 1)(2.016)]}}$$

$$A_e = \frac{(1 + 0.0293\varphi)^2 RT_e\left[5.773 + 2(\varphi - 1)\right]}{gp_e m\left[3.773(28.016) + 2(18.016) + 2(\varphi - 1)(2.016)\right]}$$

For complete expansion of combustion products, the exhaust-nozzle pressure ratio p_e/p_c was assumed equal to the ambient static pressure divided by the total pressure behind a normal shock at the flight Mach number with $\gamma = 1.4$ (ref. 5). Normal-shock pressure ratios obtained in this manner are nearly equal to those computed using real-gas equilibrium values (ref. 6) for Mach numbers of 5 and 10. For Mach 20, the pressure ratio for $\gamma = 1.4$ is less than 10 percent higher than the value calculated by use of reference 6. When exhaust-nozzle-throat conditions were computed, $p_e/p_c = 1/2$ was used.

Net thrusts were calculated in the following manner:

(1) For the nacelle-type ramjet installation, the usual net thrust expression was used:

$$F_n = m + A_e(p_e - p_0) - \frac{V_0}{g}$$

which, with complete expansion, reduces to

$$F_n = m - M_0\sqrt{\frac{\gamma_0 RT_0}{29g}}$$

(2) For the submerged ramjet, it was assumed that all of the engine except the diffuser-inlet area is covered with existing wing and body structure. Therefore, $-p_0(A_e - A_0)$ was assigned to the airframe drag, and the net-thrust equation became

$$F_n = m + p_e A_e - \left(\frac{V_0}{g} + p_0 A_0\right)$$

With $p_e = p_0$,

$$F_n = m + p_0 A_e - \sqrt{\frac{\gamma_0 RT_0}{29g}}\left(M_0 + \frac{1}{\gamma_0 M_0}\right)$$

Net thrust was divided by the fuel-air ratio to yield fuel specific impulse, similar to the rocket performance index:

$$I = \frac{F_n}{0.0293\varphi}$$

Thrust coefficients were computed for ramjet comparisons as follows:

$$C_F = \frac{F_n}{qA_0} = \frac{F_n}{\frac{\gamma}{2}\, p_0 M_0^2 A_0}$$

RESULTS AND DISCUSSION

Figure 1 describes the flight path for the submerged and nacelle-type ramjets. These engines were assumed to have normal-shock inlets. The solid lines show variations of ambient static and free-stream dynamic pressure with flight Mach number for a combustor pressure of 2 atmospheres. Relations for a 0.2-atmosphere combustor pressure are given by the dashed lines. The values for hypersonic glide vehicles will probably fall between these two limits.

In these analyses, the combustor pressure of 2 atmospheres (dynamic pressure of about 1.08 atm) was used. However, the results apply approximately to flight paths with other dynamic pressures. If an isothermal atmosphere were assumed, the results would transfer exactly; flight Mach number alone would determine the free-stream total energy and the exhaust-nozzle pressure ratio for complete expansion.

Hydrogen-rich equivalence ratios required to obtain combustor-exit temperatures of 1200°, 1600°, and 2000° K at flight Mach numbers from 5 to 20 are plotted in figure 2.

Figure 3 shows relations for fuel specific impulse, combustor-exit temperature, and flight Mach number. Results are given for both the submerged and nacelle-type ramjets with hydrogen temperatures of 150°, 500°, and 850° K. The impulse for the nacelle-type ramjet is $(p_0/0.0293\varphi)(A_e - A_0)$ less than the value for the submerged ramjet, since external pressure forces are included for the nacelle ramjet. Combustor-exit temperature controls the impulse at Mach numbers from 5 to about 7. For example, near Mach 6 for a given ramjet and a particular combustor-exit temperature, the curves for various fuel temperatures approach a common value, and impulse increases with increasing combustor-exit temperature (decreasing equivalence ratio). The approximate impulses for these points are as follows:

Combustor-exit temperature, °K	Approximate impulse at Mach 6, $\frac{\text{lb net thrust}}{\text{lb fuel/sec}}$	
	Submerged ramjet	Nacelle-type ramjet
1200	600	540
1600	800	700
2000	1050	900

Between Mach numbers of 15 and 20, hydrogen temperature exerts the principal influence on impulse. The effect of combustor-exit temperature is present, but is secondary:

Hydrogen temperature, $^{\circ}K$	Approximate impulse at Mach 20, $\frac{\text{lb net thrust}}{\text{lb fuel/sec}}$	
	Submerged ramjet	Nacelle-type ramjet
150	325	300
500	410	385
850	490	470

Thus, in the high flight Mach number range, impulse increases with increasing hydrogen temperature. Also, all trends indicate that higher impulses will occur at lower equivalence ratios. The lower equivalence ratios were not considered, because of the arbitrary maximum combustor temperature assigned in this analysis. In actual practice, both the fuel temperature and the combustion temperature will depend on the fuel flow, because the fuel will be used to absorb the aerodynamic heating load. Thus, as the fuel flow is decreased, the fuel and combustor temperatures will increase, and higher impulse values will result. The attempt to select an optimum is beyond the scope of this report, since such an attempt involves a total aircraft system analysis and a study of the recombination problem. A system analysis may well indicate the desirability of reducing the equivalence ratio and hence raising the arbitrary maximum combustion temperature.

Fuel-rich addition of 150°, 500°, and 850° K hydrogen to both submerged and nacelle-type ramjets gave the variations of thrust coefficient with flight Mach number and combustor-exit temperature shown in figure 4. Thrust coefficients for the nacelle-type ramjet differ from those of the submerged ramjet, because the net thrust is lower by the external force $p_0(A_e - A_0)$ assigned the nacelle-type ramjet. A true comparison of the coefficients of thrust and their interrelation with the aircraft requires a complete system analysis.

For a given flight Mach number and ramjet type, thrust coefficient increases with decreasing combustor-exit temperature (increasing equivalence ratio) and increasing hydrogen temperature. Curves for the thrust coefficient have minimums between flight Mach numbers of 10 and 13.

Figure 5 shows values of nozzle-throat area and nozzle-exit area each divided by the free-stream (diffuser-inlet) area as functions of combustor-exit temperature and flight Mach number for hydrogen temperatures of 150°, 500°, and 850° K. The results apply to both the submerged and nacelle-type ramjets. In general, for a given flight Mach number, exhaust-nozzle areas increase with decreasing combustor-exit temperature (increasing equivalence ratio) and increasing hydrogen temperature. Minimums for the ratio of nozzle-throat area to diffuser-inlet area occur between Mach numbers of 10 and 13.

It appears that the ratio of exhaust-nozzle-throat area to diffuser-inlet area for hydrogen-rich hypersonic ramjet operation would be less than 1.0. This ratio is consistent with current designs.

As discussed in the INTRODUCTION, it is highly desirable to be able to use a fixed-geometry ramjet engine. But a conventional ramjet, operating over a range of Mach numbers, is hampered by the large changes in ratio of inlet to exhaust-nozzle-throat area required to keep the engine operating on design. The consequence of mismatched inlet and exit areas is a loss in thrust either through losses in pressure ratio or loss in mass flow. The flat curves for the ratio of nozzle-throat to inlet area shown in figure 5 indicate that fixed-geometry systems may be practical for the hydrogen-rich ramjet. Also, some adjustments to the area ratio can be achieved through variations in fuel flow. In order to determine whether matching of inlet to exhaust through fuel dilution is feasible, the diffuser losses must be weighed against changes in fuel impulse and thrust. Only a complete mission analysis can establish this feasibility.

The complete-expansion engine requires very large exit to inlet area ratios at high flight speeds, as shown in figure 5. If the hydrogen-rich ramjet engine is to be used along an acceleration path, the variation in exit area of figure 5 is not practical. Also, in general it does not pay to expand the gases in jet engines completely, since losses from external pressure forces and exhaust-nozzle weight more than offset the small incremental gains in jet thrust.

The effects of incomplete expansion are shown in the following table for the submerged ramjet using 500° K hydrogen to yield 2000° K combustion products. The values for complete expansion of exhaust gases are taken from figures 2 to 5. In the second part of the table, approximate calculated results are presented for expansion through an exhaust nozzle with an exit area 5 times greater than the diffuser-inlet area. The selection of the area ratio is only an arbitrary example of the effect of incomplete expansion and does not attempt to reflect a "best design."

Flight Mach number	Equivalence ratio	Nozzle-throat area / Diffuser-inlet area	Nozzle-exit area / Diffuser-inlet area	Impulse, lb thrust / lb fuel/sec	Thrust coefficient
Complete expansion of 2000° K combustor-exit gases					
5	3.8	1.39	6	1320	2.05
10	9.2	.99	10.2	585	1.02
15	18.4	.97	17.1	433	.98
20	32.6	1.02	26.6	418	1.23
Expansion of 2000° K combustor-exit gases through nozzle with exit area 5 times diffuser-inlet area					
5	3.8	1.39	5	1275	1.98
10	9.2	.99	5	509	.89
15	18.4	.97	5	350	.79
20	32.6	1.02	5	303	.89

The ratio of exhaust-nozzle-exit to diffuser-inlet area for incomplete expansion was 83 percent of that required for complete expansion at a Mach number of 5 and only 19 percent that of complete expansion at a Mach number of 20. Partial expansion reduced the net thrust to 96.5 and 72.5 percent of that for complete expansion at Mach numbers of 5 and 20, respectively. This reduction in net thrust produced corresponding losses in thrust coefficient and impulse.

The values of fuel specific impulse for both complete and partial expansion are much higher than the impulse of chemically fueled rockets based on the average impulse for an acceleration path from a low to a high Mach number. The comparisons should be considered preliminary, since it has not been ascertained whether sufficient heat is available from the aircraft to produce fuel temperatures in the 150° to 850° K range. As discussed previously, fuel temperature has a strong influence on impulse at Mach numbers greater than 15. Also, the dilution of the energetic stream was assumed to occur in such a manner that no dissociated products remained. Whether this can be easily accomplished or not is largely in the opinion stage.

In support of the favorable comparison, it should be noted that conservative values for diffuser pressure recovery and low combustion temperatures were used.

From a practical viewpoint, a major problem of any hydrogen-fueled jet engine is the handling of the low-density, low-temperature fluid. The specific gravity of liquid hydrogen is about 0.071; therefore, large tanks and associated gear are required. The fuel-rich ramjet and supporting equipment must be compared with more conventional systems such as rockets before their true merit is established.

CONCLUDING REMARKS

This preliminary analysis of hydrogen-rich ramjet operation indicates that fuel specific impulses equal to or better than those of rockets can be attained. The high impulses are particularly gratifying in view of the low combustor temperatures employed and simple normal-shock diffuser selected.

The coefficients of thrust, the ratios of inlet to exhaust-nozzle-throat area, the type of diffuser used, and the very high average impulses show that this system could possibly be applied to a flight path involving acceleration from low to high flight speeds.

Factors such as the influence of high hydrogen flow rates on fuel temperatures and handling problems must be assessed before the feasibility of the hydrogen-rich ramjet system is established.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, October 18, 1957

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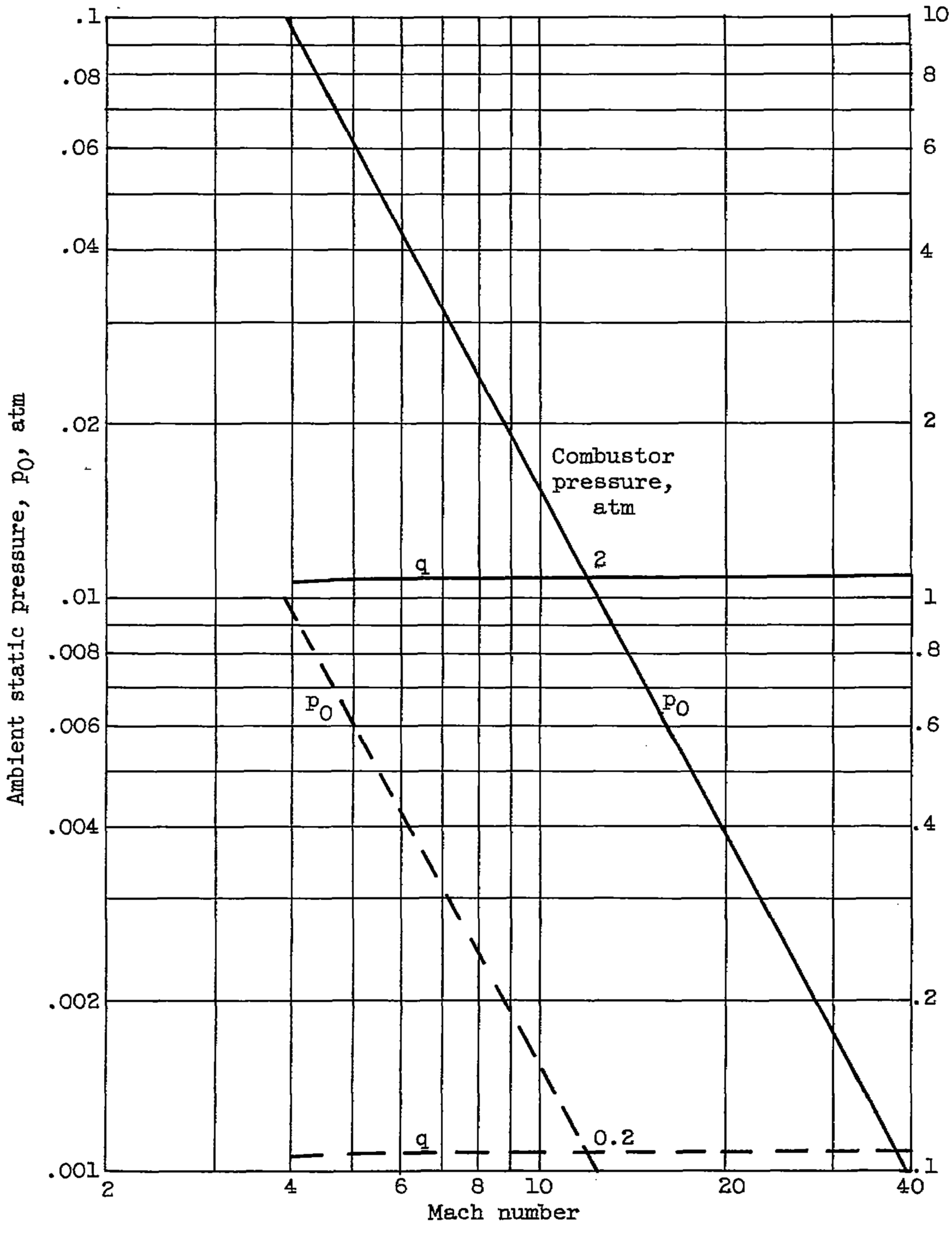

Figure 1. - Variations of ambient static and dynamic pressures with Mach number for assumed flight path.

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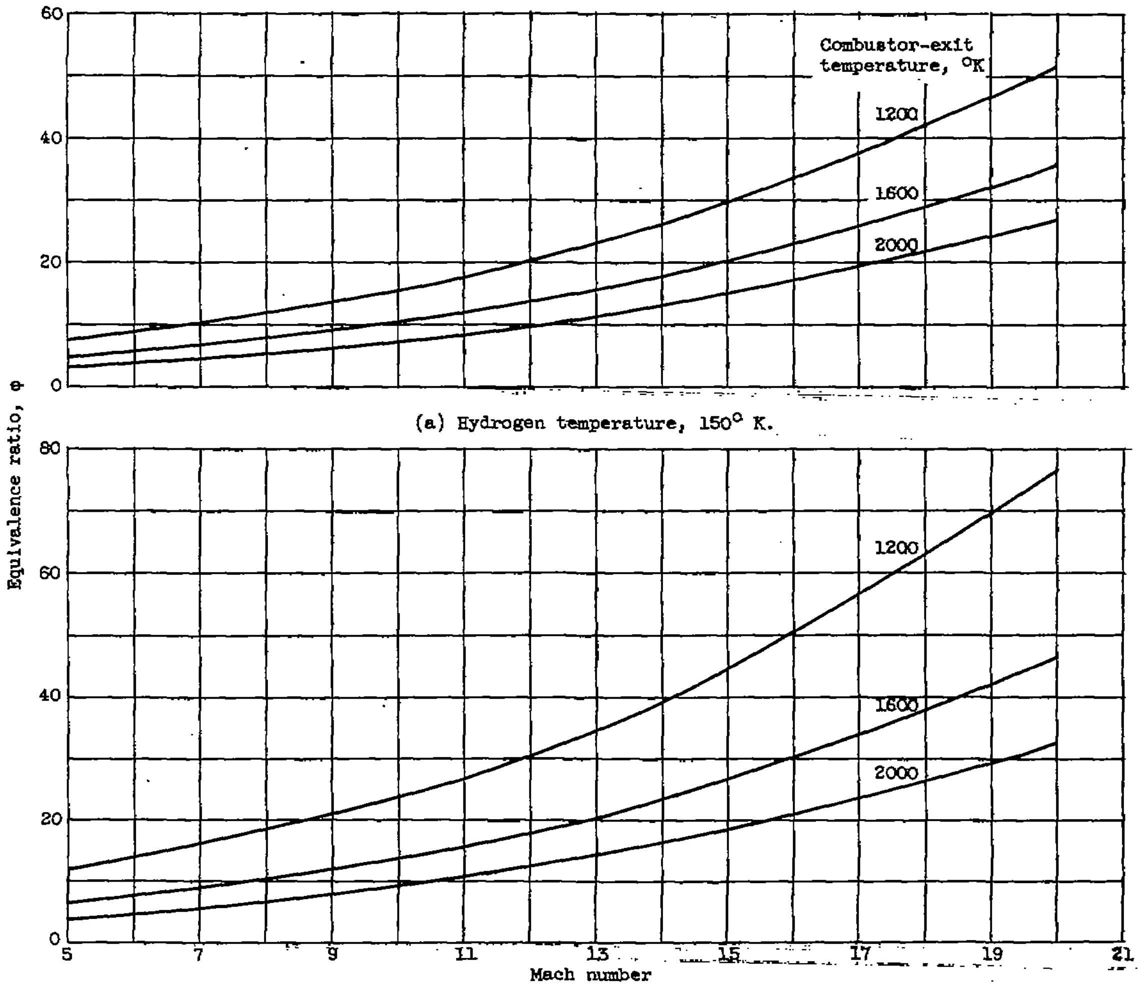
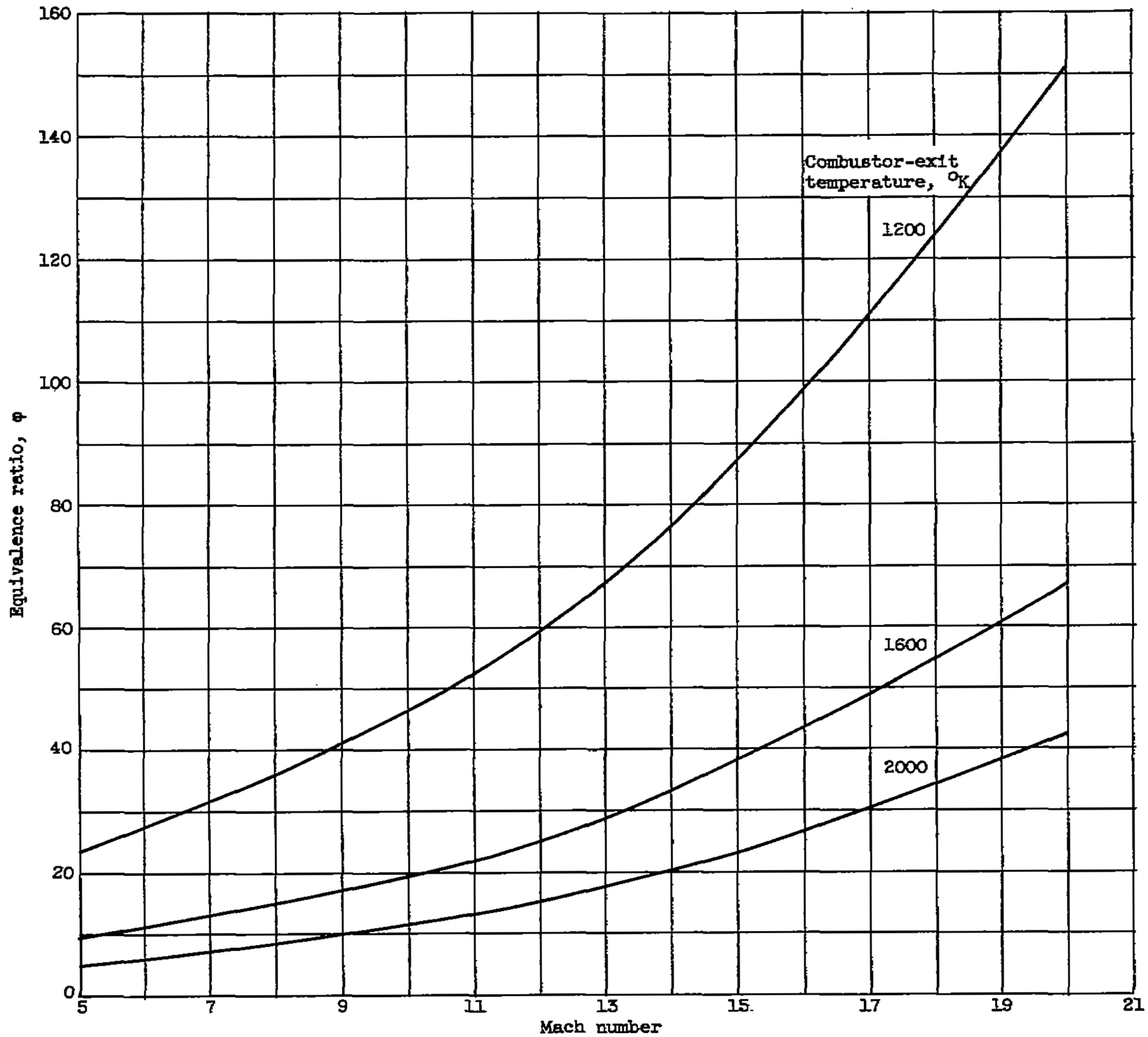


(a) Hydrogen temperature, 150° K.

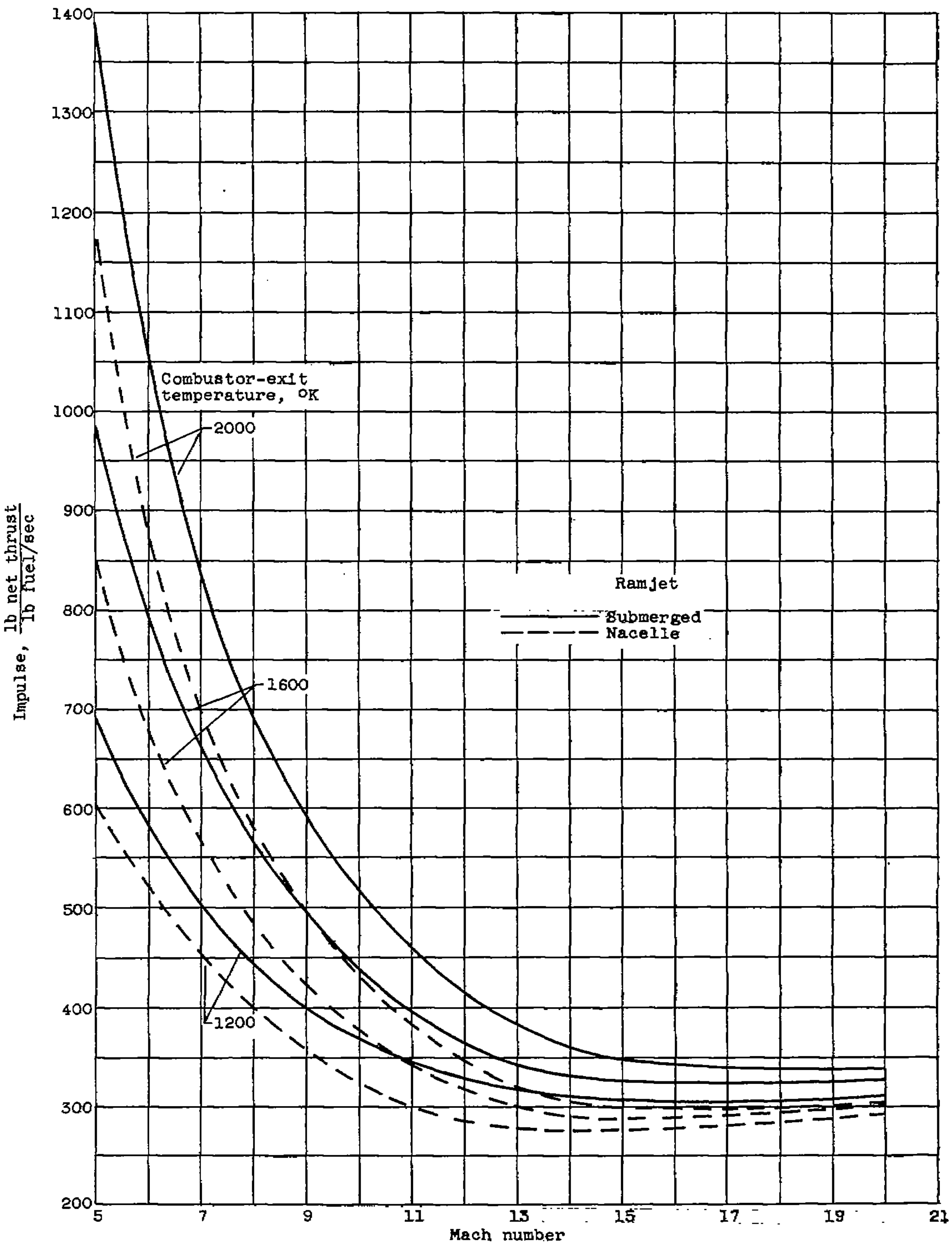
(b) Hydrogen temperature, 500° K.

Figure 2. - Hydrogen-rich equivalence ratios required for combustor-exit temperatures of 1200°, 1600°, and 2000° K at flight Mach numbers from 5 to 20.



(c) Hydrogen temperature, 850° K.

Figure 2. - Concluded. Hydrogen-rich equivalence ratios required for combustor-exit temperatures of 1200°, 1600°, and 2000° K at flight Mach numbers from 5 to 20.



(a) Hydrogen temperature, 150° K.

Figure 3. - Relation of impulse, combustor-exit temperature, and flight Mach number for hydrogen-rich operation of submerged and nacelle ramjets.

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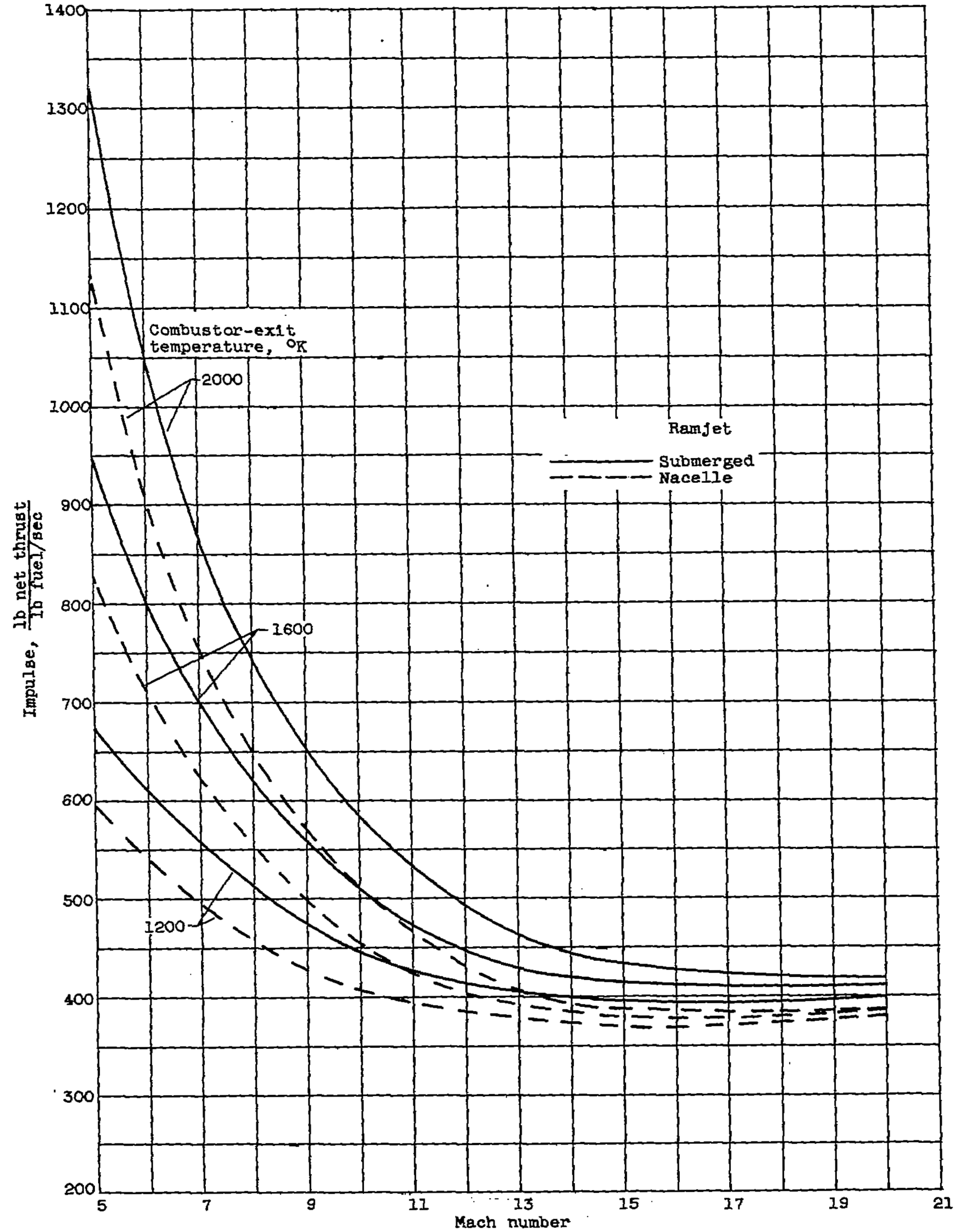
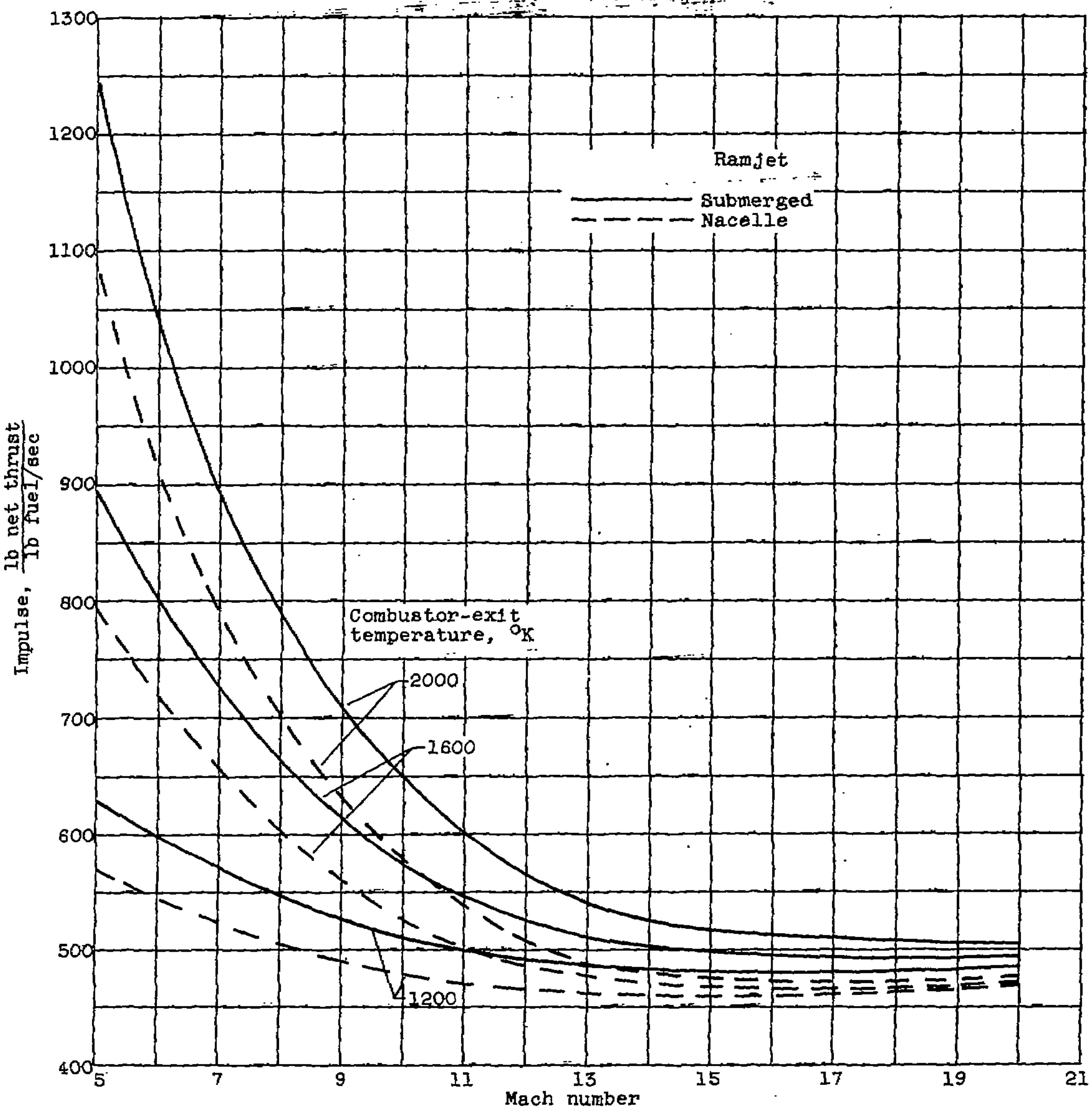


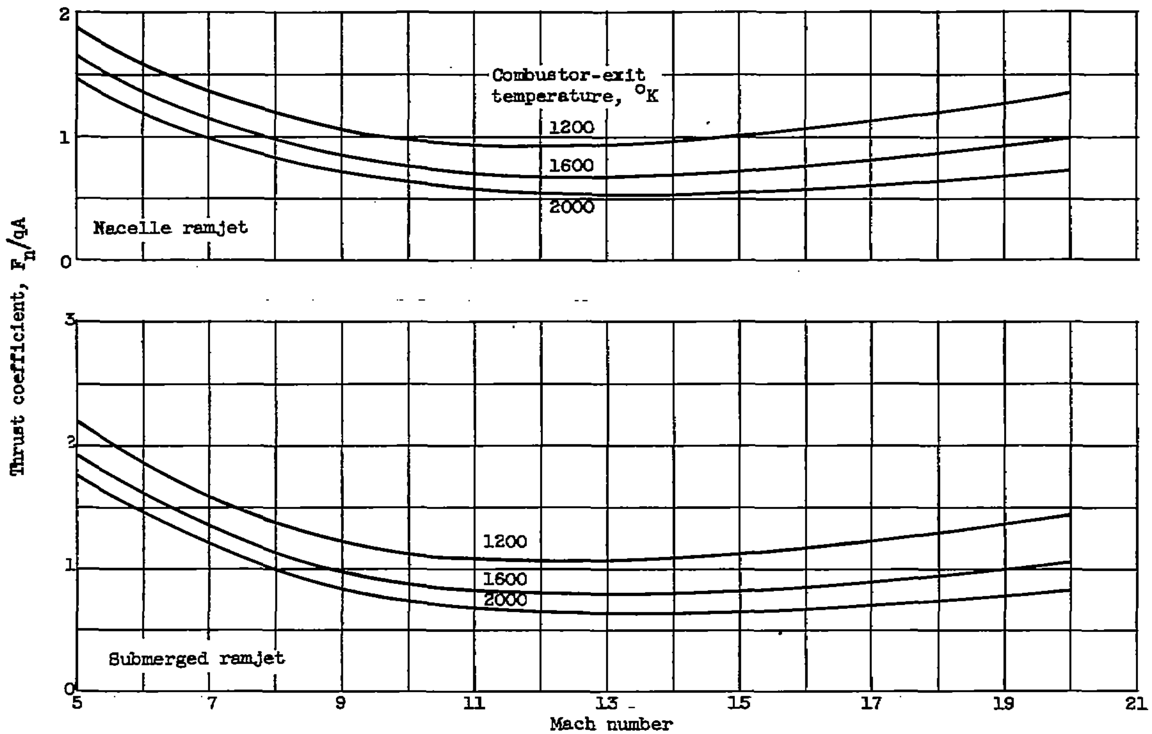
(b) Hydrogen temperature, 500° K.

Figure 3. - Continued. Relation of impulse, combustor-exit temperature, and flight Mach number for hydrogen-rich operation of submerged and nacelle ramjets.



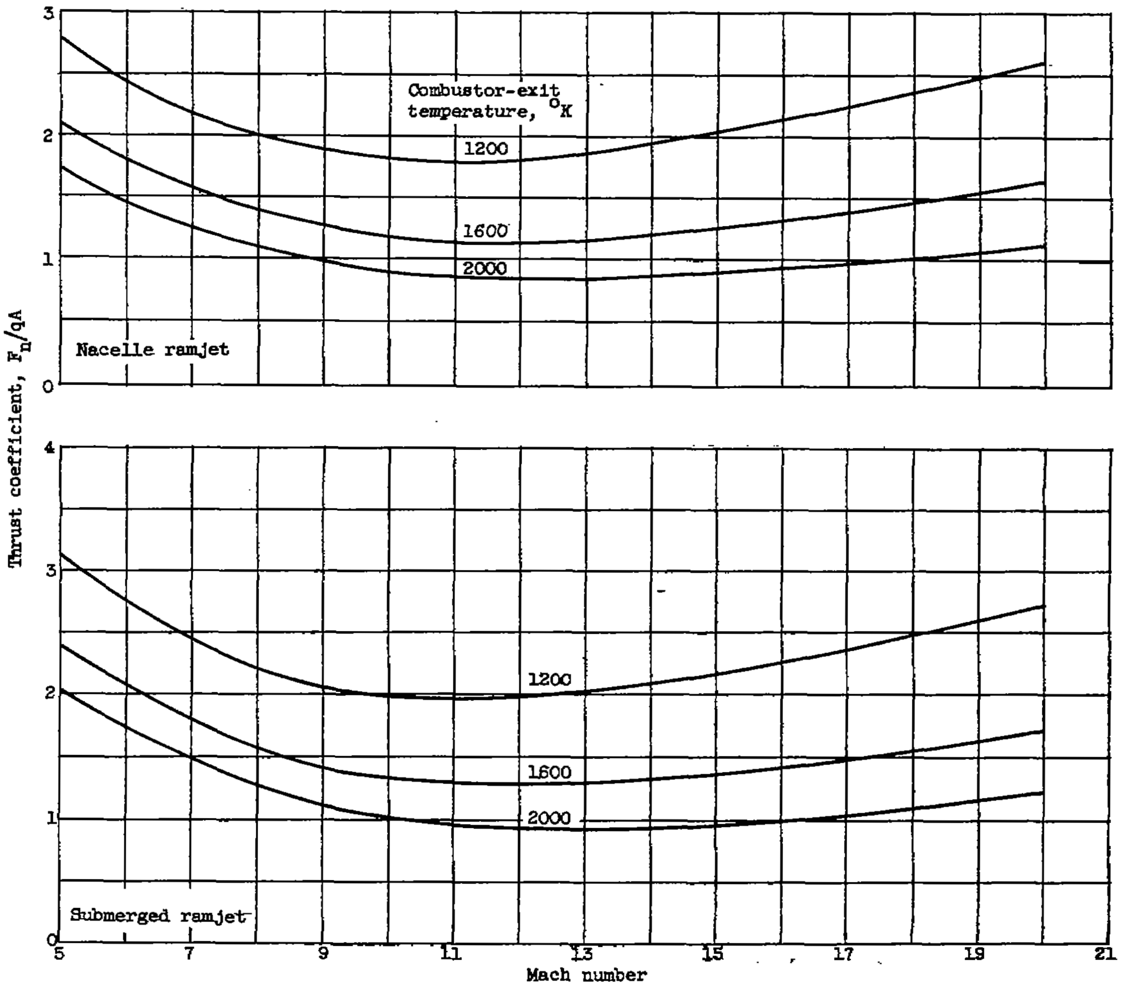
(c) Hydrogen temperature, 850° K.

Figure 3. - Concluded. Relation of impulse, combustor-exit temperature, and flight Mach number for hydrogen-rich operation of submerged and nacelle ramjets.



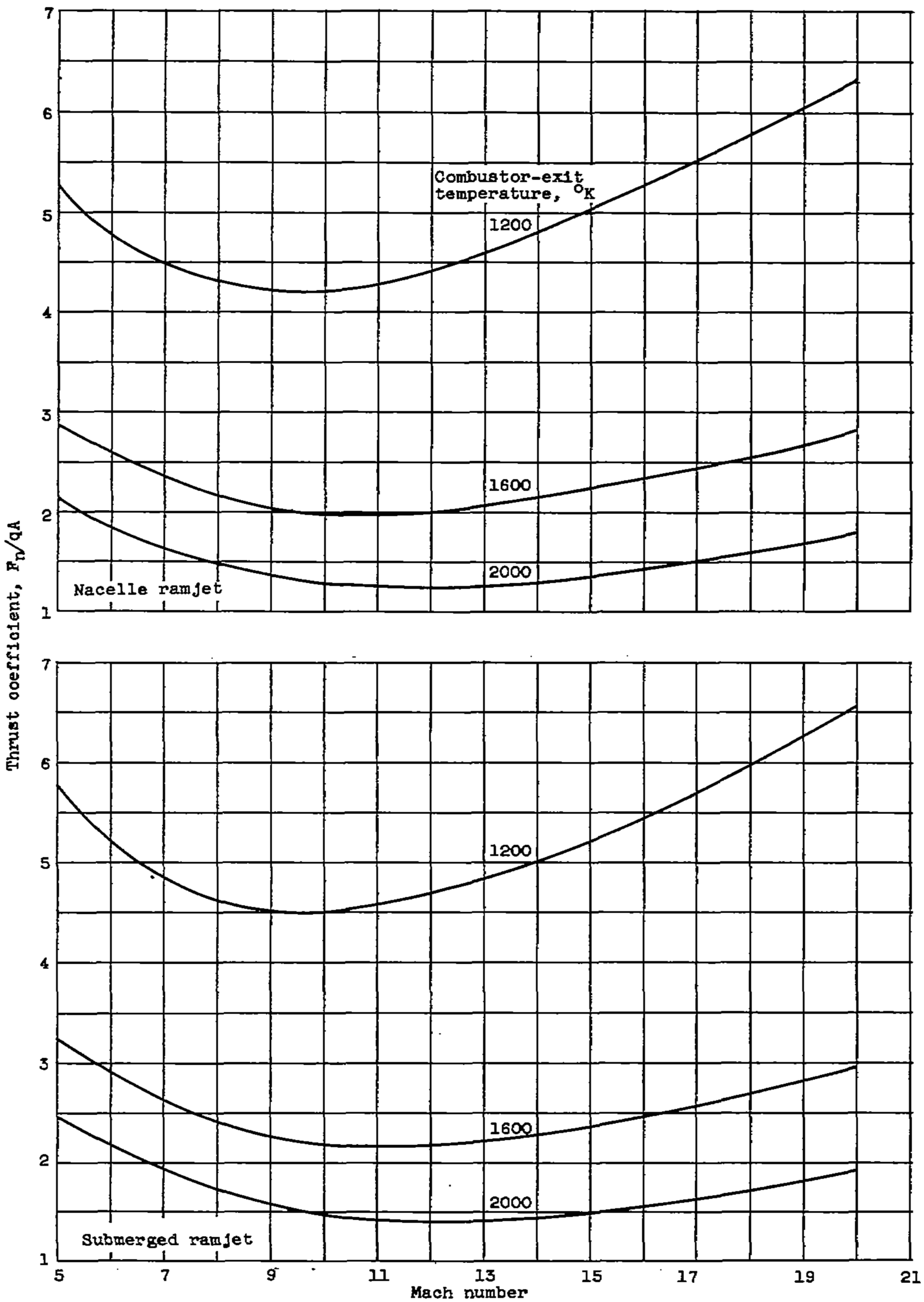
(a) Hydrogen temperature, 150° K.

Figure 4. - Variations of thrust coefficient with combustor-exit temperature and flight Mach number for hydrogen-rich operation of submerged and nacelle ramjets.



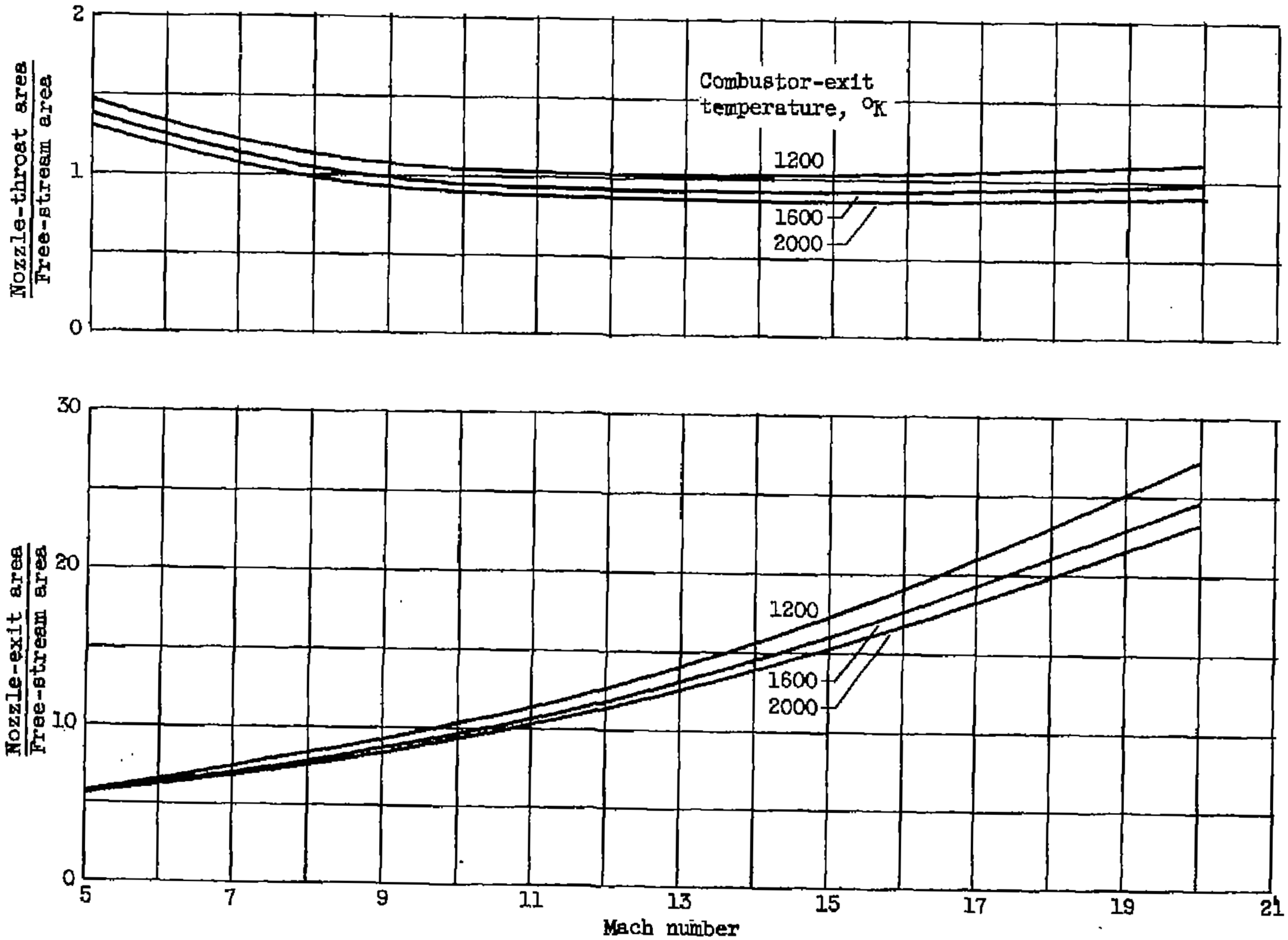
(b) Hydrogen temperature, 500° K.

Figure 4. - Continued. Variations of thrust coefficient with combustor-exit temperature and flight Mach number for hydrogen-rich operation of submerged and nacelle ramjets.



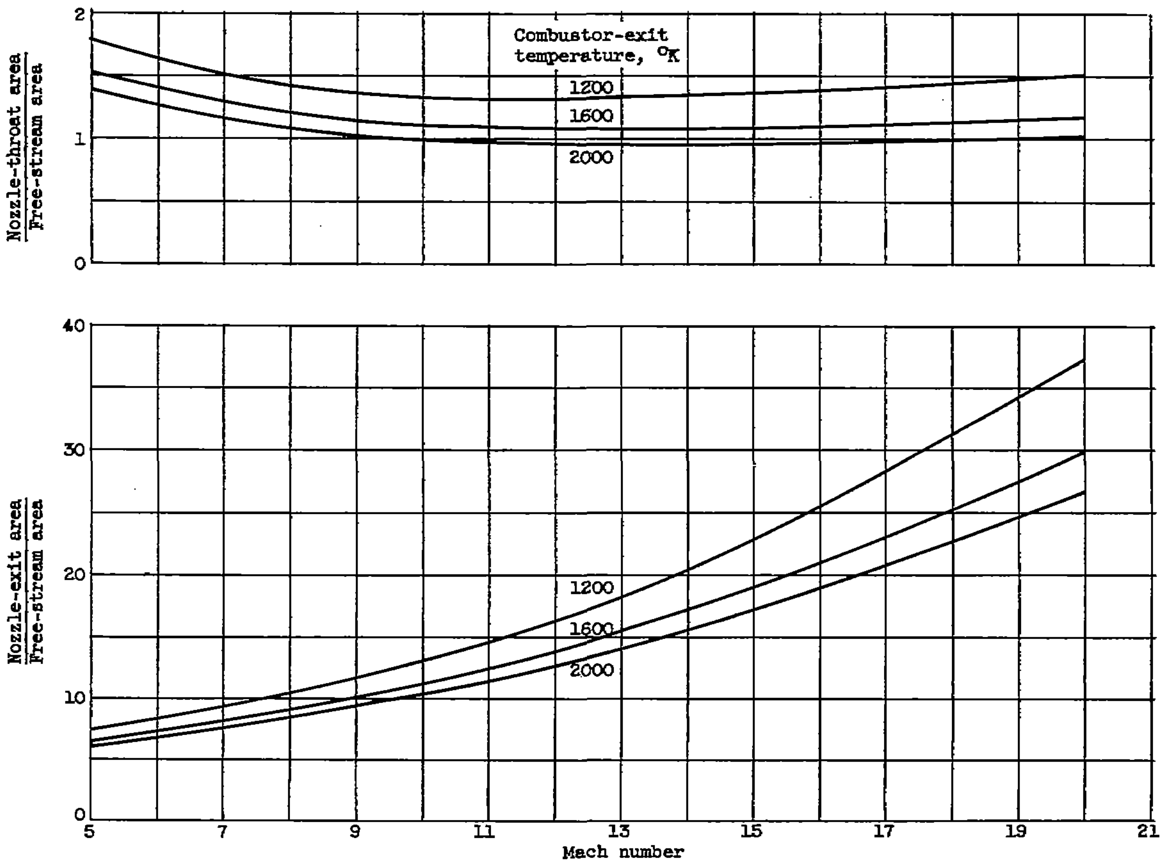
(c) Hydrogen temperature, 850° K.

Figure 4. - Concluded. Variations of thrust coefficient with combustor-exit temperature and flight Mach number for hydrogen-rich operation of submerged and nacelle ramjets.



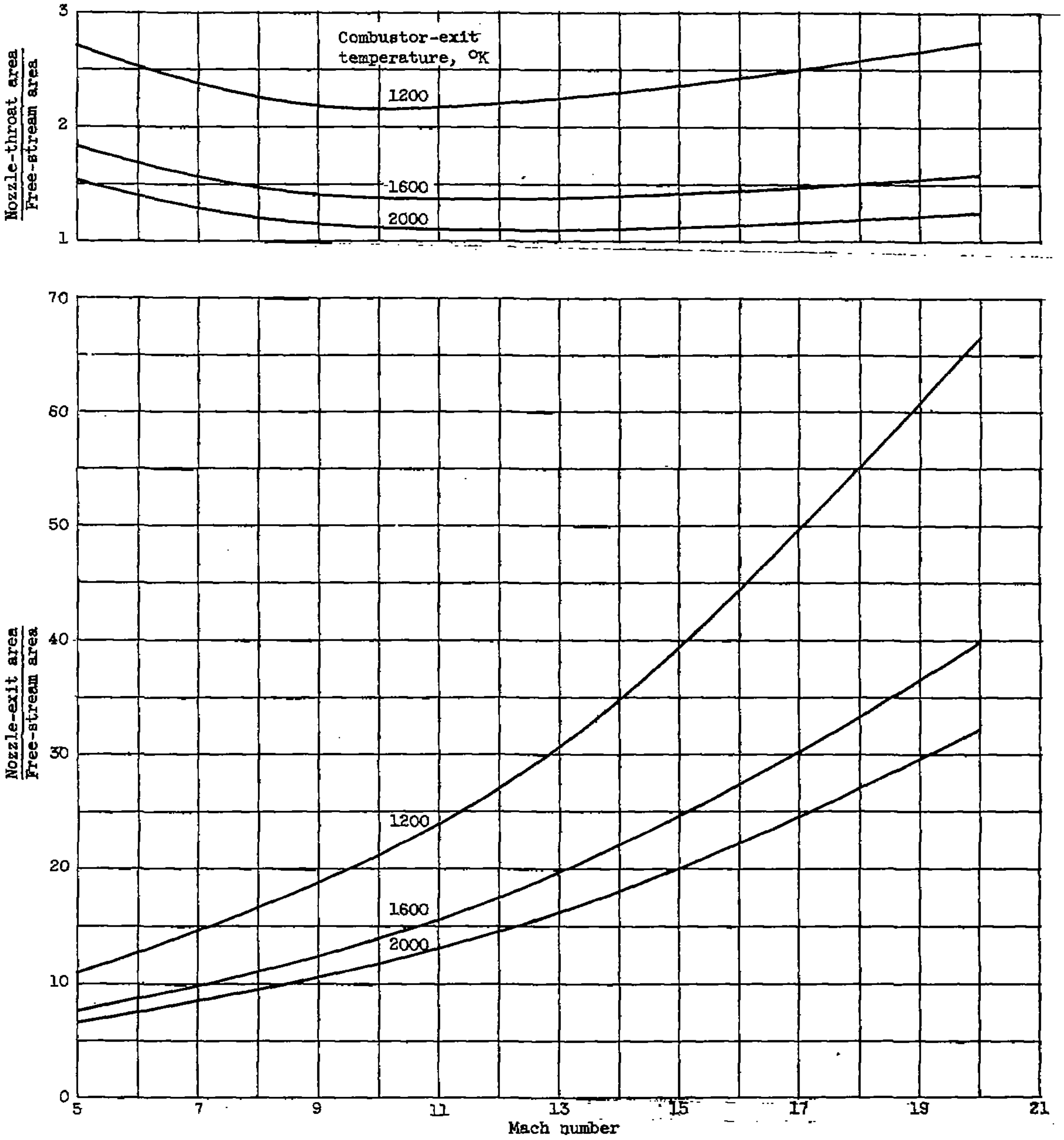
(a) Hydrogen temperature, 150° K.

Figure 5. - Variations of ratios of exhaust-nozzle-exit and -throat areas to free-stream (diffuser-inlet) area with combustor-exit temperature and flight Mach number for hydrogen-rich ramjet operation.



(b) Hydrogen temperature, 500° K.

Figure 5. - Continued. Variations of ratios of exhaust-nozzle-exit and -throat areas to free-stream (diffuser-inlet) area with combustor-exit temperature and flight Mach number for hydrogen-rich ramjet operation.



(c) Hydrogen temperature, 850° K.

Figure 5. - Concluded. Variations of ratios of exhaust-nozzle-exit and -throat areas to free-stream (diffuser-inlet) area with combustor-exit temperature and flight Mach number for hydrogen-rich ramjet operation.